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No. 17

PROJECT SQUID

CLASSIFIED SUPPLEMENT TO

SEMI-ANNUAL PROGRESS REPORT

April 1, 1954

PRINCETON UNIVERSITY

Project SQUID is a cooperative program of basic research in Jet Propulsion. It is supported jointly by the United States Army, Navy, and Air Force and is administered by the Office of Naval Research through contract N6-ori-105, Task Order 111 NR-098-038.

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CLASSIFIED SUPPLEMENT TO

SEMI-ANNUAL PROGRESS REPORT

PROJECT SQUID

A COOPERATIVE PROGRAM
OF FUNDAMENTAL RESEARCH
AS RELATED TO JET PROPULSION
FOR THE
OFFICE OF NAVAL RESEARCH, DEPARTMENT OF THE NAVY
OFFICE OF SCIENTIFIC RESEARCH, DEPARTMENT OF THE AIR FORCE
OFFICE OF ORDNANCE RESEARCH, DEPARTMENT OF THE ARMY

This Report covers the classified work accomplished during the period 1 October 1953 to 31 March 1954 by prime and sub-contractor under contract number N6-ori-105, Task Order III, NR-098-038

JAMES FORRESTAL RESEARCH CENTER
PRINCETON, NEW JERSEY

April 1, 1954

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Semi-Annual Progress Report — April 1, 1954

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(continued from unclassified section)

MIXING AND COMBUSTION PROCESSES IN RAM ROCKETS AND OTHER AFTERBURNERS

Princeton University - Phase 7

J. V. Charyk, Phase Leader
I. Glassman, J. E. Scott, Jr.

Introduction

A ram rocket power plant provides an excellent arrangement for the investigation of the mixing and burning of two parallel gas streams, one subsonic and the other supersonic, with large differences in initial stagnation temperature. The use of monopropellant rocket fuels permits one to cover a wide range of secondary energy release values and, in some instances to control afterburning so that the direct effect of energy release on the mixing process can be determined. Further, the exhaust products of methylacetylene monofuel contain a large portion of solid carbon particles; consequently the effects of a solid phase, and possible long reaction times compared to aerodynamic times, can be conveniently studied.

Discussion

Quantitative information on the mixing and burning processes occurring in a ram rocket is now being obtained in an arrangement consisting of a monopropellant rocket in an afterburner. The particular afterburner being used is so constructed that it is sectionally cooled and has static pressure taps every two inches axially along its length. Additional openings have been provided for entrance of a pneumatic traversing probe in order to permit velocity, temperature, and concentration profiles to be taken. The air is supplied to the burner by a system of centrifugal blowers. Details of construction and preliminary results have been recorded previously (1).

Figure 1 exemplifies a characteristic static pressure distribution of an experimental test with both mixing and burning of the air and rocket exhaust streams. The mixing and burning distributions obtained appear to consist of three integral parts. The first, closest to the rocket, suggests that initially there is very little mixing of the supersonic rocket exhaust jet and air; consequently, little or no static pressure change. This very slight initial spreading or mixing of a supersonic jet has been predicted analytically (2). In the second part the jet has dissipated and mixing takes place with perhaps a little burning. Thus, as is characteristic of the mixing of two streams in a

constant area duct, the static pressure rises. In the last part the burning of the fuel becomes the dominating factor and the characteristic pressure drop with energy release is obtained. Figure 2 represents the same operating conditions as Figure 1 but with the burning suppressed. Here one again notices a small section in which there is very little mixing, then, after the breakdown of the core of the central jet, the characteristic pressure rise due to the mixing of the rocket exhaust stream and air occurs. These curves are two of many which were obtained using ethylene oxide as the rocket propellant. Similar curves for mixing and burning and pure mixing have been obtained with methylacetylene. The burning curves with both fuels are similar to those obtained using a bipropellant rocket system in which the secondary energy release was substantially lower than that obtained with the monopropellants (3). From the static pressure distributions one can readily estimate the distance required for the completion of the mixing and burning processes. Attempts are being made to apply an analysis similar to that given by Charyk and Matthews (4) to the axial pressure data in order to calculate the rates of energy release through the burner and to determine the parameters affecting this release.

To date pure mixing runs have been carried out over air-fuel mixture ratios (R^*) of 2 to 15 for both ethylene oxide and methylacetylene. Runs with secondary combustion have covered an R^* range of 2 to 7.5 for ethylene oxide and a limited range around 7 for methylacetylene. These ranges are now being extended.

The burning ranges covered reflect to a degree the ease with which

secondary combustion can be initiated. As reported last time, in the preliminary system which induced its own air, secondary combustion was always spontaneous, except when quenched by physical means. In the final installation afterburning could be initiated only with some secondary ignition source. Evidence has lead to the belief that a new, very smooth opening propellant valve and a smooth approach flow to the burner are responsible for this change in operating conditions. It is believed that, starting with the valve previously used, some raw fuel was ejected through the rocket with the hot decomposition products. The ignition temperature of the fuel in air is substantially lower than any of the decomposition products and secondary combustion is facilitated. This argument has been substantiated since the injection of raw fuel directly into the burner proved satisfactory for secondary ignition.

When the approach section to the burner is removed and the rocket is permitted to induce its own air, the flow entering the burner separates from the wall. It would appear that the flow detaching from the wall facilities better mixing, causes a more suitable fuel-air ratio at the rocket exit, and thus creates more favorable conditions for ignition. There is some experimental evidence to support this reasoning.

In the ethylene oxide tests the injection of raw fuels sufficed for secondary ignition up to R^*'s of approximately 7.5. Although above 7.5 the flame would blow out of the burner, this is not the limiting air-fuel ratio for operation. It is characteristic of the particular piece of experimental equipment used that an increase of the air mass flow is accompanied by an increase of burner inlet velocity. The rocket or fuel

mass flow is normally kept constant. Consequently, since the blow out limit is more likely a function of the inlet velocity, the rocket mass flow will be halved to extend the R^* range of operation. Thus the rocket will operate at about 50 lbs. thrust instead of the normal 100 lbs. thrust.

The data which have been obtained can also be applied to calculating the performance of the ram rocket power plant. Shown in Figure 3 is a comparison of a theoretical and experimental air specific impulse as a function of air-fuel mixture ratio for an ethylene oxide system. The efficiency of the experimental burner over the range covered is seen to be exceptional. The efficiencies greater than 100% in the very rich region (stoichiometric for ethylene oxide is 7.8) is due probably to the fact that the calculated theoretical value may be actually higher than shown. The theoretical air specific impulse is a function of the temperature, and the estimation of a true theoretical adiabatic flame temperature in the very rich region is doubtful. As further results are obtained they will be plotted on Figure 3 to extend the complete range. A similar plot will be constructed for the data obtained on methyl-acetylene.

In the analyses discussed above there has arisen a need for theoretical specific impulse of an ethylene oxide rocket operating at various chamber pressures. In numerous places in the literature there appear conflicting impulse values, which are due partly to the use of the incorrect value of the heat of formation at 25°C of liquid ethylene oxide and partly to the equilibrium decomposition products chosen. In a

relatively low temperature reaction as in the ethylene oxide decomposition process one would expect to find carbon as a product. Experimental operations verify however that in the processes occurring in ethylene oxide rockets little or no carbon is found in the exhaust gases. It seemed advisable, therefore, in calculating the specific impulse of ethylene oxide rocket motors to choose as the products of the decomposition reactions CO, CH_4, CO_2, C_2, H_2O, H_2, C_2H_4. Using the heats of formation, equilibrium constants, enthalpies, and heats of vaporization from the latest National Bureau of Standards Circulars (5,6) and carrying out an adiabatic decomposition temperature determination for ethylene oxide, it was found that the molar quantities of CO_2, O_2, and H_2O present in the decomposition products were negligible and could be neglected. Shown in Tables I and II are the results of these calculations.

Table I

Performance of Ethylene Oxide as a Function of Pressure

Pc Atms	Tc °K	M. W.		C^* m./sec	Isp secs
20	1288	21.15	1.174	1189	159.4
30	1300	21.26	1.173	1194	168.0
40	1306	21.32	1.172	1196	173.6
60	1312	21.36	1.171	1197	180.7

Table II

Products of Decomposition Corresponding to Table I

Pc	Moles CO	Moles CH_4	Moles H_2	Moles C_2H_4
20	1.00	0.84	0.16	0.08
30	1.00	0.86	0.14	0.07
40	1.00	0.87	0.13	0.07
60	1.00	0.88	0.12	0.06

In Table I, T_c is the adiabatic temperature of the decomposition process, c^* the characteristic exhaust velocity, P_c the chamber pressure, the ratio of mean specific heats of the gas mixture, M. W. the mean molecular weight of the gas misture, and I_{sp} the specific impulse calculated assuming frozen equilibrium and one atmosphere exhaust pressure. The products of decomposition are for one mole of reactant.

During the next period the range of operation will be extended to complete the work on both ethylene oxide and methylacetylene. The analysis of the static pressure of distributions will be completed and all data reported in a final report.

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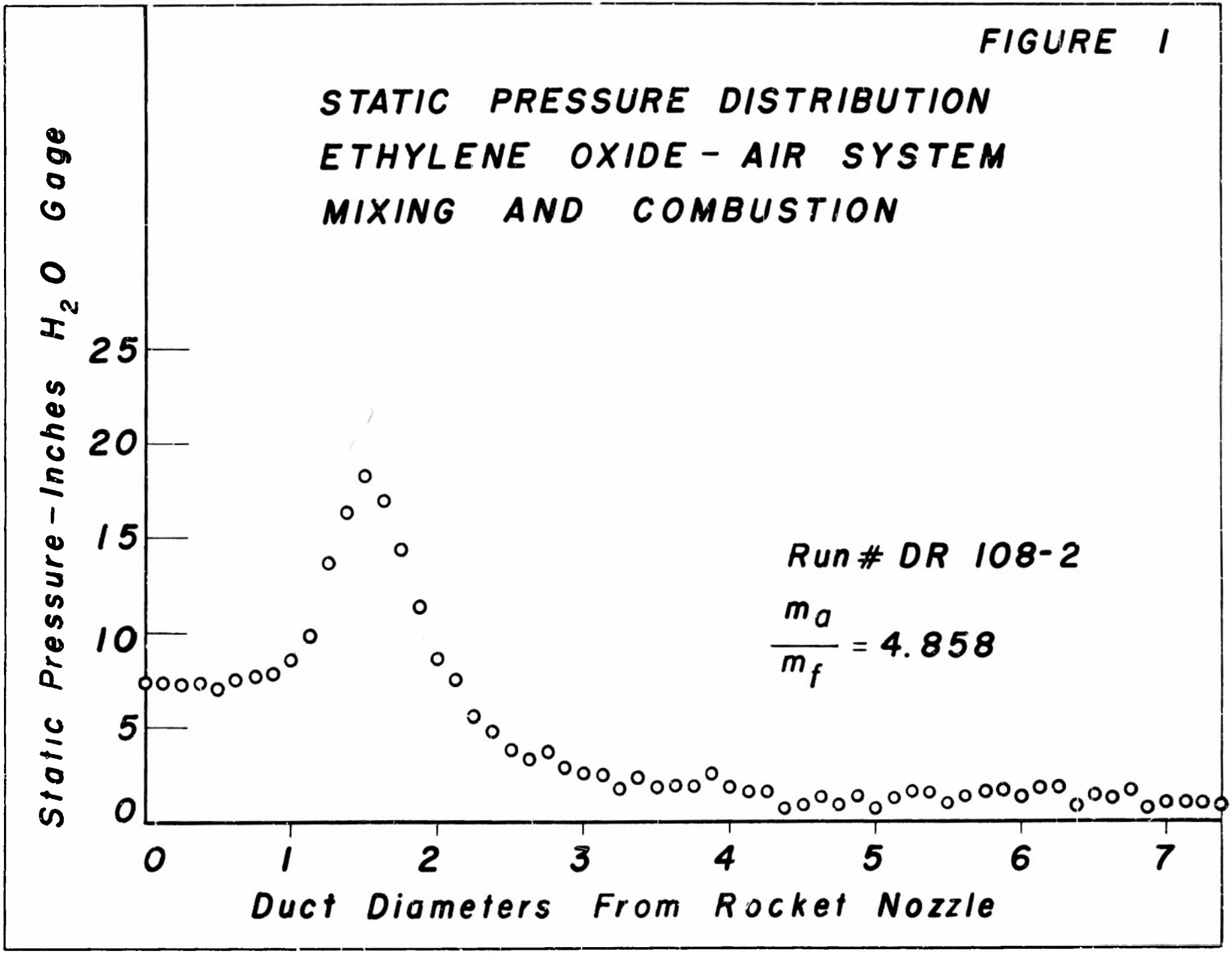
2. Warren, W. R., Jr., "The Mixing of an Axially Symmetric Compressible Jet with Quiescent Air," Princeton University Aero Engr. Lab. Report No. 243, September, 1953.
3. Charyk, J. V., Glassman, I., and John, R. R., "The Mixing and Burning of Two Concentric Fluid Streams," Fourth Symposium on Combustion, Williams and Wilkins, Baltimore, 1953.
4. Charyk, J. V., and Matthews, G. B., "Experimental Studies of Energy Release Rates in Rocket Motor Combustion Chambers," Princeton University Aero. Engr. Lab. Report No. 191, March, 1952.
5. National Bureau of Standards Circular C-461, "Selected Values of Properties of Hydrocarbons, 1947.

6. National Bureau of Standards Circular 500, "Table of Selected Values of Chemical Thermodynamic Properties, 1950.

FIGURE 1



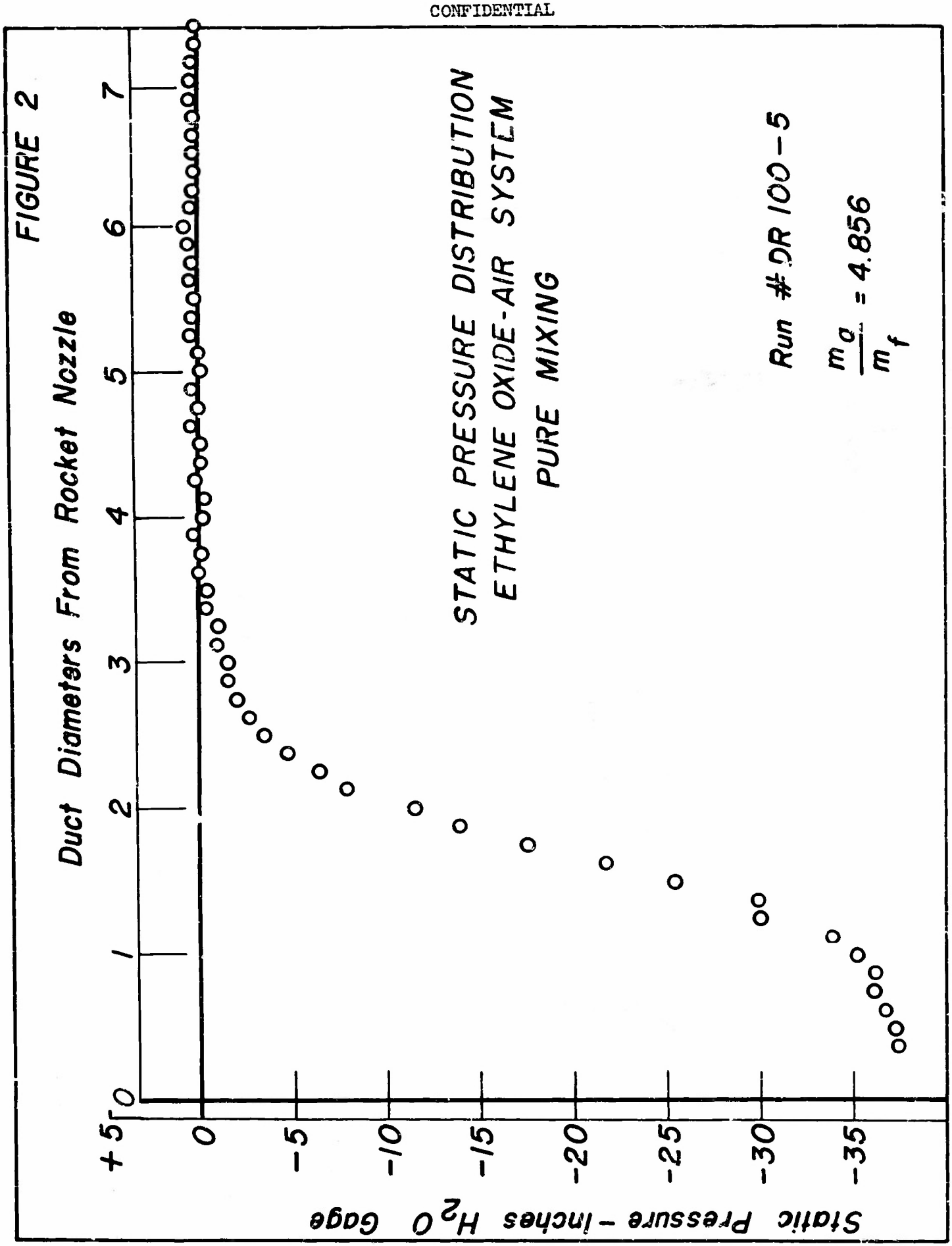


FIGURE 2
Duct Diameters From Rocket Nozzle
0
1
2
3
4
5
6
7
STATIC PRESSURE DISTRIBUTION
ETHYLENE OXIDE-AIR SYSTEM
PURE MIXING
Run # DR 100-5
$\frac{m_a}{m_f}$ = 4.856
+5
0
-5
-10
-15
-20
-25
-30
-35
Static Pressure - inches H_2O Gage

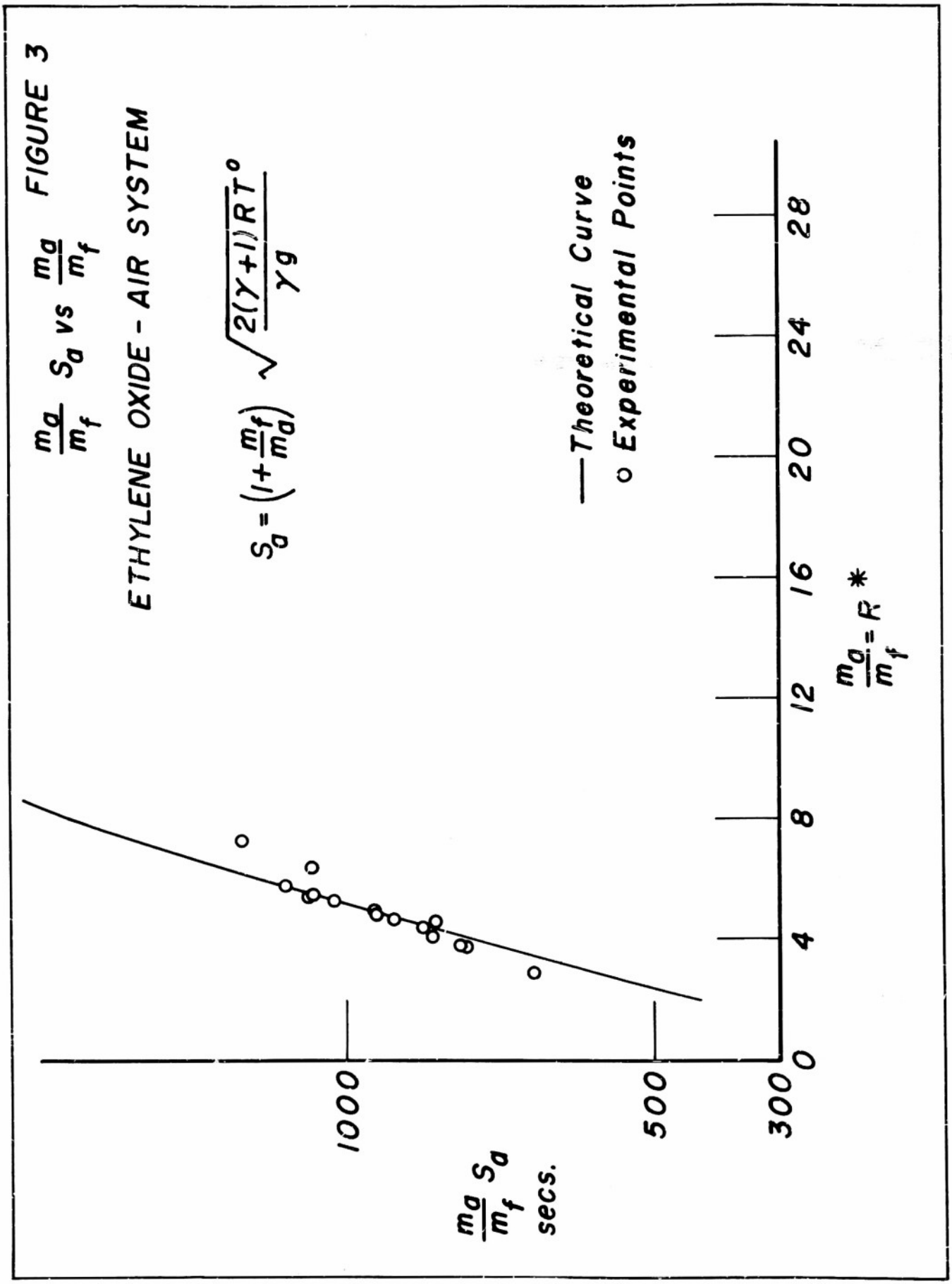


FIGURE 3
$\frac{m_a}{m_f}$ S_a vs $\frac{m_a}{m_f}$
ETHYLENE OXIDE - AIR SYSTEM
$S_a = \left(1+\frac{m_f}{m_a}\right)\sqrt{\frac{2(\gamma+1)RT^\circ}{\gamma g}}$
—Theoretical Curve
○ Experimental Points
$\frac{m_a}{m_f}$ secs.
1000
500
300
0
4
8
12
16
20
24
28
$\frac{m_a}{m_f} = R^*$

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